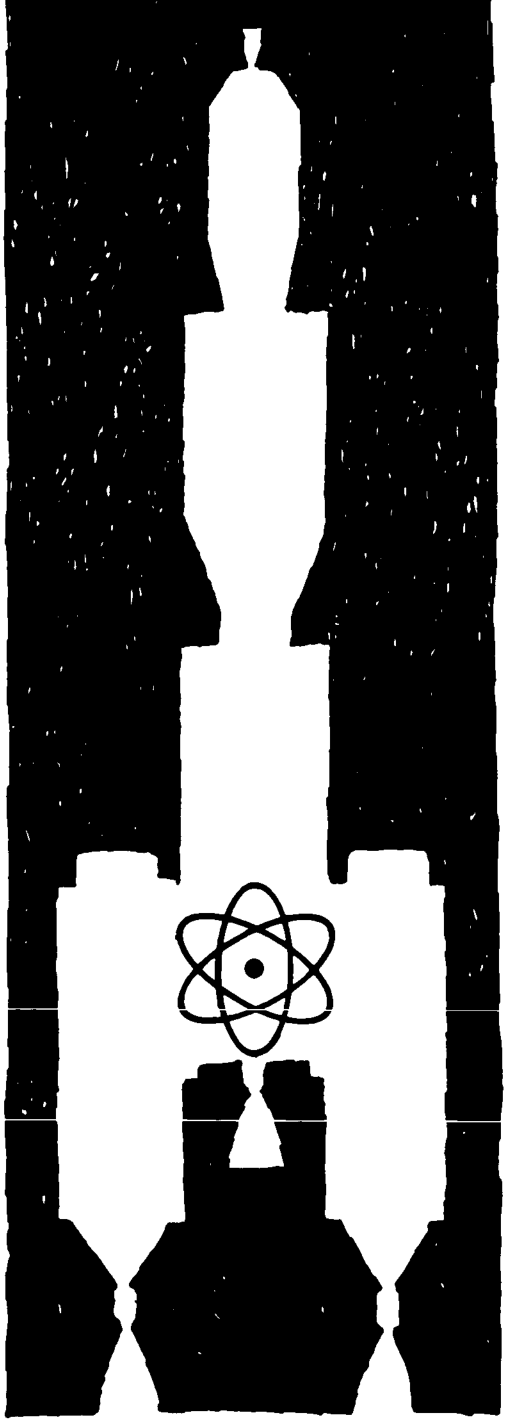
MISSION ORIENTED ADVANCED NUCLEAR SYSTEM PARAMETERS STUDY



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FOR

GEORGE C. MARSHALL SPACE FLIGHT CENTER

BY

TRW SPACE TECHNOLOGY LABORATORIES

THOMPSON RAMO WOOLDRIDGE INC.

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MISSION ORIENTED ADVANCED NUCLEAR SYSTEM PARAMETERS STUDY

Final Report

Volume VI

Research and Technology Implications Report

for

George C. Marshall Space Flight Center
National Aeronautics and Space Administration

by

TRW Space Technology Laboratories
Redondo Beach, California

Volume VI Research and Technology Implications Report

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FOREWORD

This volume, which is one of a set of nine volumes, describes in part the studies, analyses, and results that were accomplished under Contract NAS8-5371, Mission Oriented Advanced Nuclear System Parameters Study, for George C. Marshall Space Flight Center, Huntsville, Alabama. This work was performed during the period from April 1963 to March 1965 and covers Phases I, II, and III of the subject contract.

This final report has been organized into nine separate volumes on the basis of contractual requirements and to provide a useful and manageable set of documents. The volumes in this set are:

Volume I	Summary Technical Report
Volume II	Detailed Technical Report; Mission and Vehicle Analysis
Volume III	Parametric Mission Performance Data
Volume IV	Detailed Technical Report; Nuclear Rocket Engine Analysis
Volume V	Parametric Nuclear Rocket Engine Analysis Results
Volume VI	Research and Technology Implications Report
Volume VII	Computer Program Documentation; Mission Optimization Program; Planetary Stopover and Swingby Missions
Volume VIII	Computer Program Documentation; Mission Optimization Program; Planetary Flyby Mission
Volume IX	Computer Program Documentation; Nuclear Rocket Engine Optimization Program

Volumes I, II and IV include the details of the study approach and basic guidelines, the analytic techniques developed, the analyses performed, the results obtained and an evaluation of these results together with specific conclusions and recommendations. Volumes III and V contain parametric mission, vehicle, and engine data and results primarily in graphical form. These data present the interrelationships existing among the parameters that define the mission, vehicle, and engine. Volume VI delineates those areas of research and technology wherein further efforts would be desirable based on the results of the study. Volumes VII through IX describe the computer programs developed and utilized during the study and present instructions and test cases to enable operation of the programs.

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ABSTRACT

A discussion is presented of the areas of research and technology wherein further efforts would be desirable based on the results of a comprehensive, parametric lunar and interplanetary mission analysis. The areas include the development of future space technology, supplementary research on alternative operational and system techniques, and the specification of vehicle weight requirements and subsystem performance characteristics.

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ILLUSTRATIONS

I INTRODUCTION

This final report delineates those areas of research and technology wherein further efforts would be desirable based on the results obtained during Phases I, II, and III of the Mission Oriented Advanced Nuclear System Parameters Study performed by TRW Space Technology Laboratories for the George C. Marshall Space Flight Center.

The overall objective of this study was to provide the information necessary for the selection of the design and operating parameters for an optimum nuclear rocket engine, or engines for interplanetary missions in the 1975 to 1990 time period. This was accomplished by (1) analyzing in detail the relationships existing among the engine design parameters and constraints and the engine performance, (2) formulating the vehicle system requirements and criteria for manned interplanetary flights, and (3) determining and evaluating the mission performance characteristics that can be expected from vehicles propelled by the advanced nuclear engines.

In order to conduct these engine, vehicle, and mission analyses for the years 1975 to 1990, it was necessary to establish many technological assumptions and guidelines for this time period. In some cases, these criteria were based on extrapolations of current technology or on the results of related NASA and industry studies; in other cases significant parameters were varied over a range of values within which the future state-of-the-arts and system requirements would exist. Since these assumptions and guidelines circumscribed the vehicle's systems performance and weight requirements, it naturally followed that the mission performance characteristics for a given engine would be influenced by the choice of these assumptions and guidelines.

In reviewing the results and conclusions obtained in this study (Vols. I, II, III, IV, and V) there emerged a number of technological or system areas the definitions of which bear heavily on the selection and ultimate performance potential of the advanced nuclear engine. These areas fall into three categories. The first concerns the space technology that will be available in the future. This includes the structural, temperature, and fabrication limitations of various materials, the development of systems and techniques for performing certain mission phases or operations, and the determination or prediction of space environments.

The second area involves alternative operational or system techniques that were discovered or defined during the study but due to the lack of time and budget were only analyzed in a cursory manner. Some of the preliminary analysis results gave evidence that these alternative techniques could improve the overall mission performance characteristics and therefore, influence the operational criteria and selection of the advanced nuclear engine. As a result, supplementary research in these areas appears warranted.

Finally, there is a third category of items that pertains to discrete vehicle weight requirements (such as payloads) or individual subsystem performance characteristics (such as attitude control and midcourse corrections). Collectively, these requirements and characteristics are translated into a significant portion of the overall vehicle weight. Therefore, their specification can have a decided effect on the overall mission capabilities of the advance nuclear engine.

The remainder of this volume discusses the areas of desirable future effort in terms of these three categories, i.e., Future Technology, Supplementary Research, and Mission Requirements.

II FUTURE TECHNOLOGY

ENGINE CONSTRAINTS

The nuclear rocket engine parametric analysis results presented in Volume V showed that the attainable engine performance is a strong function of the engine design constraints. Therefore, the selection of the best nuclear engine design for interplanetary missions is highly dependent on the engine state-of-the-art design constraints. At present, very limited experimental information is available which enables these limitations to be well defined. Thus, the experimental determination of the engine design constraints is of utmost importance in the engine selection process. Of primary importance is the determination of the fuel element temperature limitation since the attainable specific impulse is ultimately determined by this temperature. For each 100 degree increase in peak fuel temperature and, thus, in exit gas temperature, the 1982 manned Mars stopover mission could be performed with a savings of approximately 40,000 pounds in initial vehicle weight, a significant reduction for a two million pound vehicle.

The achievable engine performance is also a strong function of the power density limitations imposed on the fuel element by thermal stress (fuel element web temperature rise) and the manufacturing and structural limitations which determine the minimum allowable fuel element web thickness. For a 50 percent increase in reactor power density, a vehicle weight decrease of 30,000 pounds can be realized. Additional constraints which must be evaluated are the maximum allowable nozzle wall temperature and the maximum allowable core pressure drop. If the allowable nozzle wall temperature can be increased by 100° R, the increased engine performance produces a vehicle weight savings of 10,000 pounds for the 1982 Mars mission. The significant reduction in vehicle weight which can be achieved by higher core pressure drops is shown in Figure 1. At a chamber temperature of 4500° R, a vehicle weight savings of 16,000 pounds is possible for each additional 50 psi of core pressure drop.

Finally the engine selection is very dependent on the engine firing times. Firing times of less than 45 minutes are required for engines delivering the optimum thrust for the missions investigated. The theoretical and experimental determination of the engine design constraints is required before the best engine design can be specified. The PHOEBUS I tests should be used to greatly expand the available knowledge on the design constraints of beryllium-reflected, graphite-moderated nuclear rocket reactors.

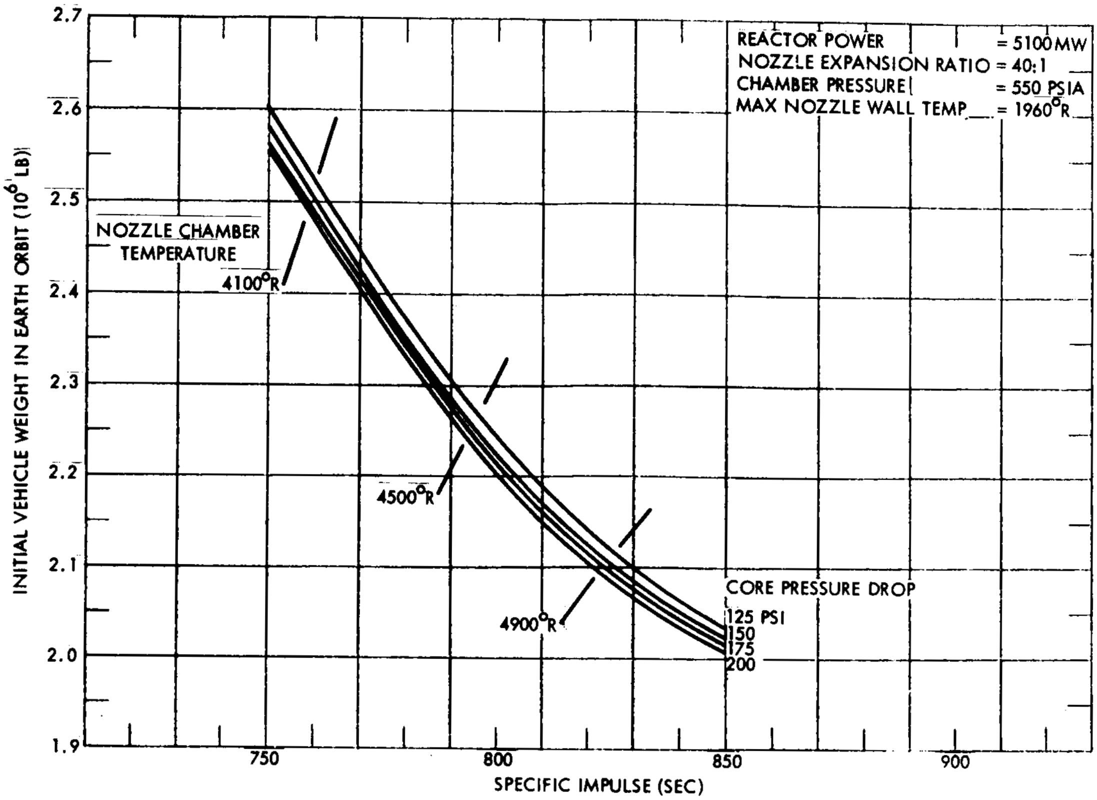

Figure 1 Vehicle Weight vs Specific Impulse and Core Pressure Drop

LAUNCH WEIGHT - AERODYNAMIC BRAKING - AND MISSION YEAR

The study revealed that of all the non-nuclear engine oriented areas related to future technology, three areas exert the greatest influence on the overall mission performance characteristics. These are (1) the total gross weight that would be available in Earth orbit for any given mission, (2) the aerodynamic Earth braking capability to be developed by this time period, and (3) the year in which the first manned Mars stopover mission could take place.

More or less educated estimates were made to define what could be regarded as reasonable, future state-of-the-art advancements in these areas for the years 1975 to 1990. These estimates were (1) a maximum of 3 to 5 million pounds in Earth orbit, (2) Earth aerodynamic braking for entry velocities up to 15 km per sec, and (3) 1982 as the year of the first manned Mars stopover mission.

Despite the selection of these advanced criteria for evaluation purposes, the study analyses were not confined to these estimates. Rather, aerodynamic braking from Apollo technology (parabolic entry velocity) to entry velocities greater than 20 km per sec were analyzed for all launch opportunities from 1975 to 1990.

The results of these analyses showed that the initial vehicle weight varied by factors of two to three (1.5 to 4-5 million pounds) for a mission performed in the most favorable year (1986) to the least favorable (1978 or 1992). Similar extreme variations in vehicle weight requirements also result for any given year for the extreme possibilities of Earth aerodynamic braking capabilities. These effects are shown in Figs. 2 and 3.

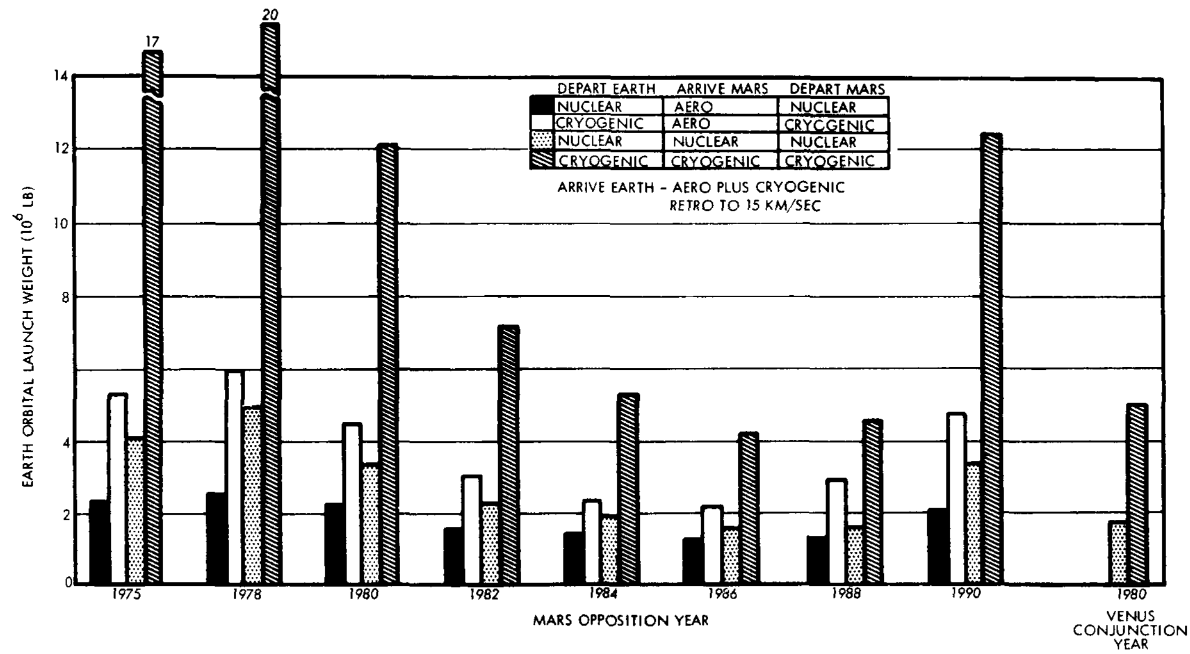

Figure 2 Variation of Vehicle Weight with Mission Year

Since the optimum thrust levels for the manned Mars vehicles are primarily a function of the vehicle weight, the optimum thrust requirements can vary significantly for the different launch opportunities (see Fig. 4) as well as for variations in the Earth aerodynamic braking capability for any given year. Conversely, for an engine of established thrust level, the payload capability will be reduced or the performance requirements of other systems, such as aerodynamic braking, will be increased if the gross vehicle weight increases.

It is evident that the explicit determination of one or more of the aforementioned three areas of future technology can greatly assist in the accurate pinpointing of the nuclear engine thrust requirements as well as accurately establishing the requirements and limitations of other currently undefined vehicle technologies. For example, if the maximum weight available in Earth orbit for any given mission was determined to be say 2.5 million pounds for this time period, then the first mission opportunity would occur in 1982 for which an Earth aerodynamic braking capability of 15 km per sec entry velocity would be required. The range of optimum nuclear engine thrusts would be dictated by the requirements for the opposition years from 1982 through 1988. This thrust range would be approximately between 100,000 and 200,000 pounds; the selection of a thrust level within this range would be a function of possible burning time limitations and operational tradeoffs of the number of engines in the Earth depart stage.

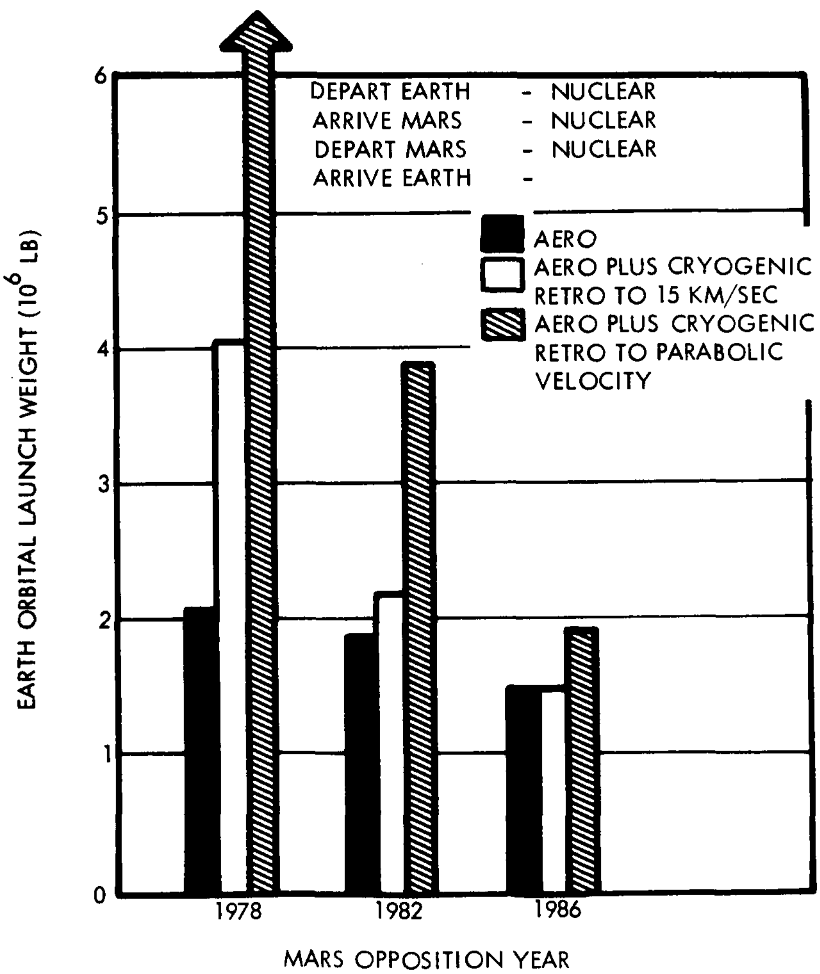

Figure 3 Variation of Vehicle Weight with Earth Aerodynamic Braking Capability

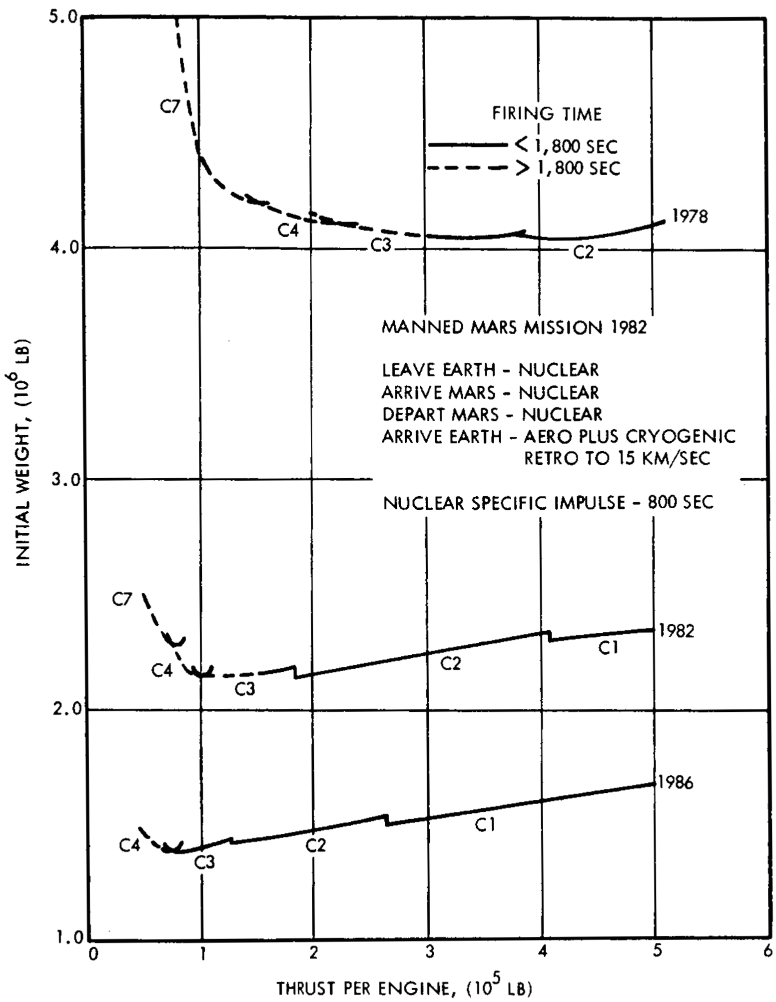

Figure 4 Nuclear Engine Thrust Requirements

If the gross orbital weight and/or the aerodynamic braking capabilities could be determined as a function of time, the mission performance characteristics could be more accurately estimated and the overall system and program requirements more accurately established for the manned interplanetary space program.

As mentioned previously, estimates of these future capabilities were made during the study for evaluation purposes. Any estimates made at this time are for technological areas in which no major or significant research and development programs currently exist. Implementation of significant programs in these areas, therefore, appears extremely desirable if accurate assessments and reliable planning for manned interplanetary missions are near term objectives.

The determination of the maximum available orbital weight involves further investigations into and development of orbital rendezvous, assembly, and check-out techniques and equipment; decisions on future booster development programs and booster availability; and the planning and development of the associated Earth launch facility requirements.

The determination of the Earth aerodynamic braking capability requires an intensified research and development program to advance the state-of-the-art past the currently planned Apollo technology.

The establishment of the target year of the initial Mars stopover flight involves a large number of interrelated elements and cannot be established per se. In order to establish this date, data are required on the orbital weight and aerodynamic braking capabilities, as well as, to varying degrees, on the objectives and results of other vehicle system research and developmental programs. This information must be coupled with the establishment of national space goals and decisions on the future committment of the nation's resources.

PROPELLANT TANK DESIGN

Another vehicle system that greatly influences the initial vehicle weight is the inert or jettison tank weight. Variations in the weight of the hydrogen tanks can occur due to structural and subsystem requirements, required micro-meteoroid protection, and techniques of long term, cryogenic storage.

Figure 5 shows the effect on vehicle weight for variations in tank jettison weight as a function of mission year. The tank mass fractions (ratio of total usable propellant to total gross tank weight less the nuclear engine) vary approximately linearly between case numbers. The nuclear stage scaling laws employed give an average mass fraction of 0.88 for case number 1, 0.84 for case number 2, and 0.80 for case number 3.

The figure shows that approximately 20 percent more vehicle weight is required for the 1986 mission for a vehicle whose propellant tank mass fractions are decreased by about 10 percent (mass fraction case number 1 to case number 3). This same decrease in propellant tank mass fractions increases the vehicle weight by over 40 percent in 1982 and by almost 150 percent in 1978.

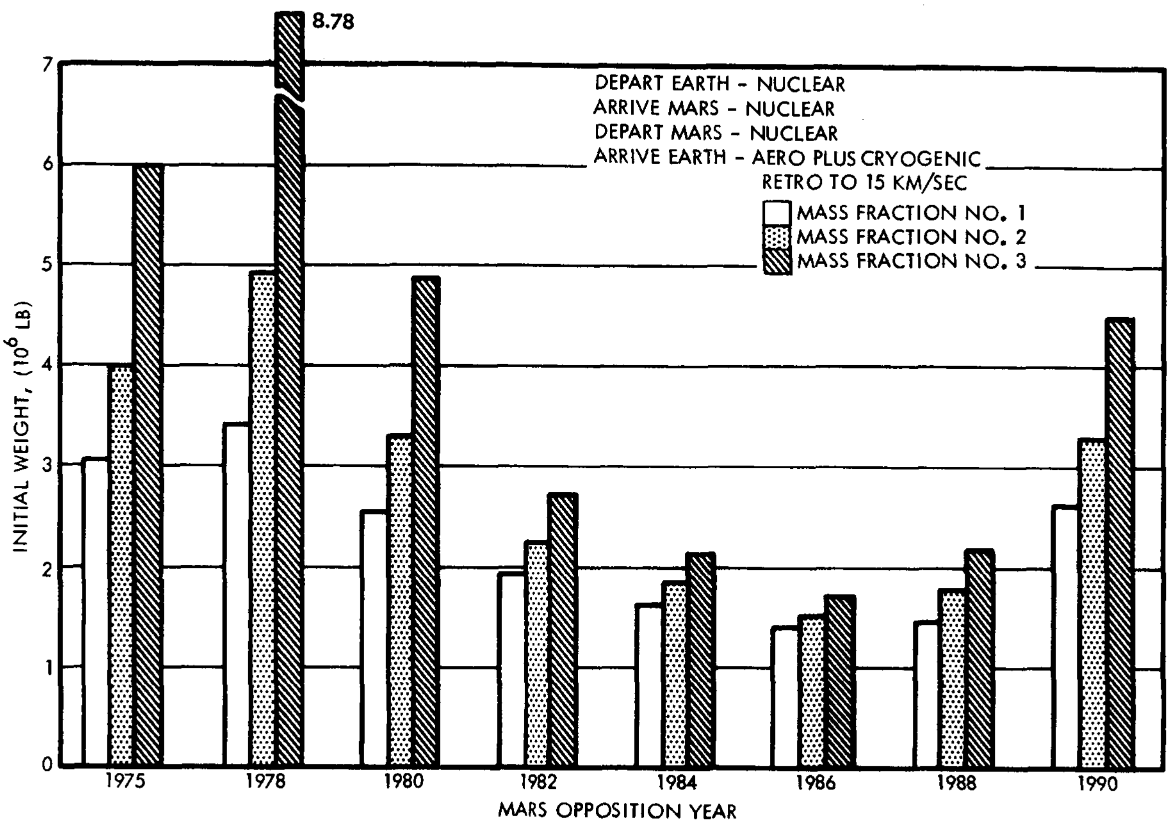

Figure 5 Vehicle Weight Variation as Function of Mass Fraction and Mission Year

Continuing and extensive research studies and test programs would be desirable to generate accurate design criteria upon which the structural design and subsystem requirements of the tanks could be based and accurate scaling laws developed. The technical areas encompassed by this research should include material technology, orbital docking equipment and assembly techniques, vehicle interstage and tank clustering requirements, and problems associated with propellant storage and feed systems under zero gravity and space environments. The determination of the micrometeoroid protection and cryogenic propellant storage requirements requires investigations into the existing space environments as well as the development of protection and storage techniques that are efficiently compatible with the tank structure, and exert a minimum penalty on the overall vehicle weight.

SOLAR FLARE SHIELDING

The mission evaluation results showed that an increase in solar flare shielding of 7,000 pounds would increase the overall vehicle weight by approximately 110,000 pounds for a 1978 Mars mission, 70,000 pounds for a 1982 mission, and 50,000 pounds for a 1986 mission. Hence, any change in solar flare shielding weight can have a considerable influence on the mission characteristics. The development of solar flare prediction techniques, and the efficient integration of the required shielding into the spacecraft design are required to reduce the weight of the solar flare shield as far as possible.

III SUPPLEMENTARY RESEARCH

ENGINE PARAMETERS

The engine performance was influenced significantly by the engine design variables. Additional study is required to fully delineate the best combination of engine parameters which produces the highest performance engine. The most influential engine design parameter was found to be nozzle expansion ratio. As shown in Fig. 6, an increase in nozzle expansion ratio from 40:1 to 120:1 resulted in a seven percent reduction in initial vehicle weight. Therefore, it is necessary to study in more detail, the trade-off between nozzle expansion ratio, interstage weight, and vehicle configuration to determine the influence of expansion ratio on vehicle performance for various missions. Such studies should define the optimum nozzle expansion ratio.

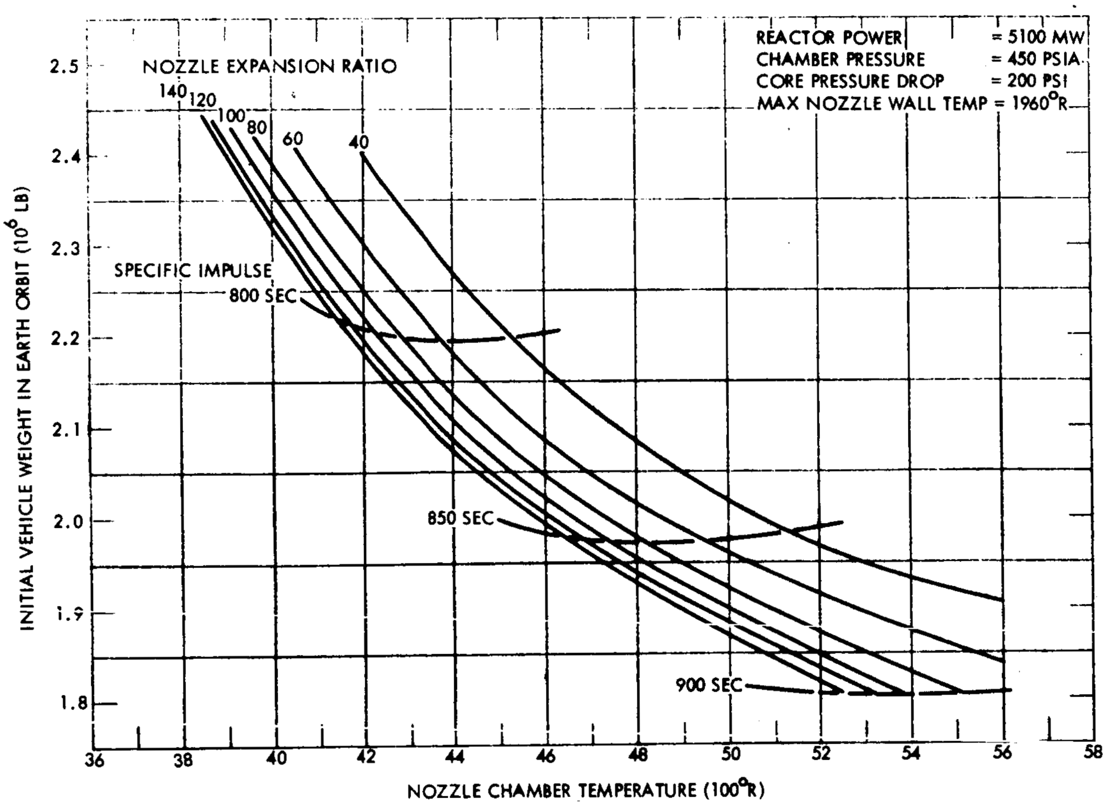

Figure 6 Vehicle Weight vs Nozzle Chamber Temperature and Expansion Ratio

Nozzle chamber pressure of bleed cycle engines is shown in Fig. 7 to have a significant effect on engine and vehicle performance. Vehicle weight additions greater than 5 percent can result from the improper selection of nozzle chamber pressure at high specific impulses. The sensitivity of engine performance to chamber pressure is closely related to the turbopump characteristics. Additional study is required to determine the trade-off between chamber pressure, turbopump weight, and efficiency to determine the combination which leads to the highest overall engine performance. This information would provide valuable information on the performance advantages or disadvantages of various types of pumps and of multistage turbines.

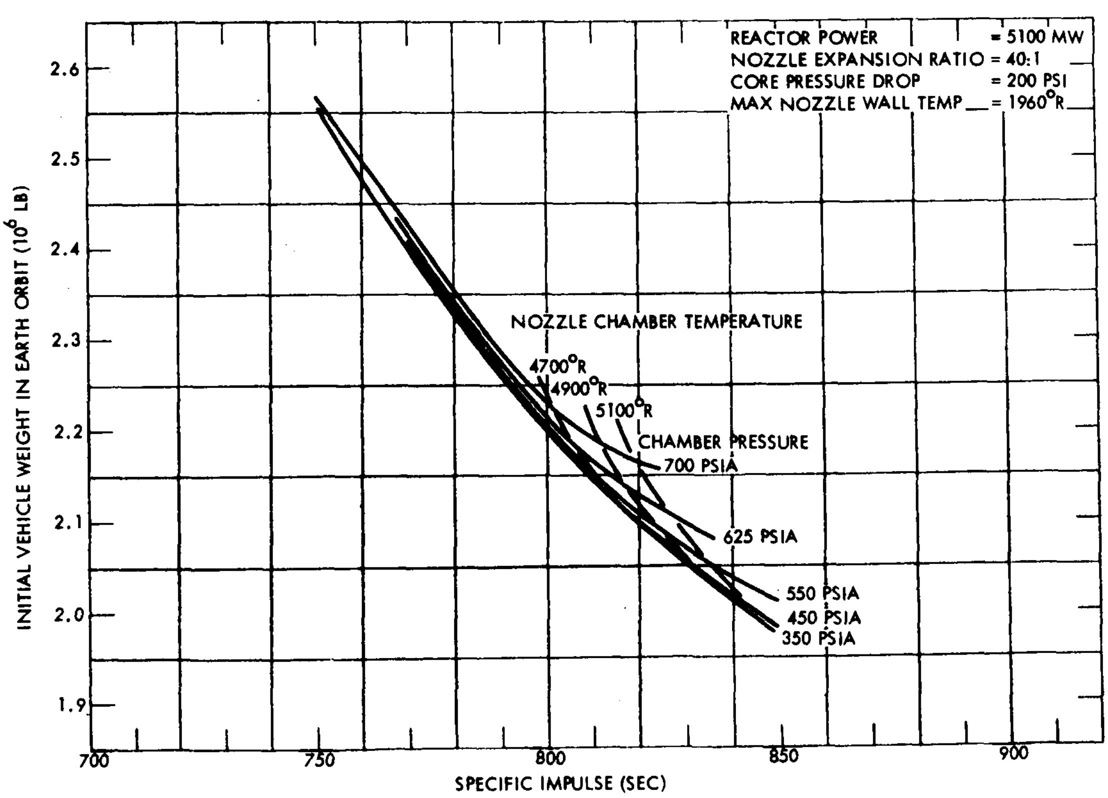

Figure 7 Vehicle Weight vs Specific Impulse and Nozzle Chamber Pressure

During the course of this study, it was evident that further investigation was necessary of the problems associated with engine clustering. This investigation should determine the weight penalties associated with attaching single and multiple engines to single and clustered tanks. Shielding, thrust, and clustering structure weight penalties need to be evaluated. In addition, the nuclear interactions of clustered engines should be assessed.

Inherent with any nuclear rocket engine is an associated radiation field. Further study is required to determine whether propellant heating or radiation damage to engine components limits the radiation fields surrounding the reactor. Once such criteria are established, a comprehensive investigation should be initiated to analyze the trade-offs between propellant tank weight, interstage weight, pump weight, shield weight, and thrust structure weight to determine the optimum combination of these parameters.

Since the topping cycle appears to offer a performance advantage over comparable bleed cycle engines, the performance of bleed and topping cycle engines should be evaluated and compared for various missions. Results of this study indicated that for certain conditions, a topping cycle engine could reduce the vehicle weight for a Mars mission by about 3.5 percent over that achievable by a comparable bleed cycle engine. A more detailed comparison would provide valuable information to determine if performance incentives could be realized by the development of a nuclear rocket engine utilizing a topping turbine cycle.

Another area of investigation requiring further study is the design of auxiliary components. This investigation would provide necessary information for accurately estimating weights of these components. Also, the design and test of flight weight components is required to accurately determine the weight of a flight-type nuclear rocket engine.

MARS AERODYNAMIC BRAKING

The use of aerodynamic braking at Mars can result in comparatively large vehicle weight reductions. Use of this braking mode reduces the vehicle weight for a nuclear propelled vehicle by approximately 50 percent in 1978, 30 percent in 1982, and 20 percent in 1986 (See Fig. 2). If the shield weight required for Mars aerodynamic braking could be reduced by 25 percent from the nominal value assumed, the initial vehicle weights would be further reduced by 8 to 12 percent.

These results show the large weight advantages to be gained through the use of this mode. Therefore, detailed, rigorous studies should be conducted to determine the technical feasibility, development requirements, and accurate scaling laws for Mars aerodynamic braking.

VENUS SWINGBY MISSIONS

The study results indicated that some of the extremes in vehicle weight variations due to the unfavorable years or high Earth arrival velocities could be eliminated and the overall vehicle weight requirements reduced by resorting to Venus swingby trajectories. Reductions in vehicle weight of over 20 percent were found to be possible for some of the cases investigated. The investigations made during the study were by no means exhaustive and future effort in this area is certainly desirable in order to determine the ultimate potential of both gravity and powered turn, Venus swingby modes.

CLUSTERED ENGINES

Although the clustering of nuclear engines for the depart Earth stage was fully explored during the study, only a limited amount of mission analysis was performed for vehicle configurations that employed clustered engines for the Mars braking and depart phases. In addition, the scope of the study did not allow a comprehensive investigation of the use of two different size nuclear engines for any given mission.

Limited results obtained late in the study indicated that for some favorable mission years significant vehicle weight savings were possible if nuclear engines were clustered on the Mars stages or if two different nuclear engines were used. Additional analysis effort in this area appears warranted.

MODULAR STAGE DESIGN

All propellant tanks in this study were based on continuous function scaling laws. That is, all propellant tanks for any given vehicle were sized for the optimum amount of propellant required for each stage. A limitation was placed on the maximum capacity of any tank and if this were exceeded, an additional tank was added to the tank cluster. Although, for any given design criteria, this method of sizing tanks will produce the minimum weight vehicle, in actuality, it is doubtful if the many different size tanks which would be required could practically be designed, fabricated, and tested.

Therefore, additional study analysis should be performed for selected missions to determine the trade-offs available and the vehicle weight penalties associated with the use of a series of propellant tanks of fixed but graduated sizes. The number of different sizes in the series should be varied parametrically.

LAUNCH WINDOWS AND HOLDS

Additional analysis should be performed to determine the overall vehicle and stage weight requirements for parametric variations of Earth launch holds. The results of this analysis will not only indicate the amount of additional weight required in Earth orbit but will also provide information as to what constitutes reasonable launch windows and the sensitivity of these launch windows to system and performance variations.

IV MISSION REQUIREMENTS

There exist a sizable number of vehicle weight requirements that are a function of the mission subsystem performance characteristics. These must all be specified in a study of this type in order to completely define the vehicle. Individually, many of these requirements have a small effect on the mission characteristics but collectively, they can influence the engine performance requirements to a marked degree. Hence, the more accurately these items can be specified, the more valid and intransient will be the results and conclusions.

These mission and subsystem performance requirements are listed below.

- Life support expendables
- Midcourse correction velocities
- Attitude control
- Orbit adjustment
- Mars descent module
- Mars ascent module
- Mission module
- Earth recovered payload
- Arbitrary payload expenditures

Some of the values specified in this study for these items were based on an extensive amount of prior analysis and study while others represented preliminary estimates. In any case, a comprehensive review of these requirements should be undertaken in the near future with emphasis placed upon predicting and incorporating future associated state-of-the-arts and integrating subsystems into complete preliminary vehicle designs.